



RESEARCH MEMORANDUM

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SECTIONS AND 40° SWEEPBACK

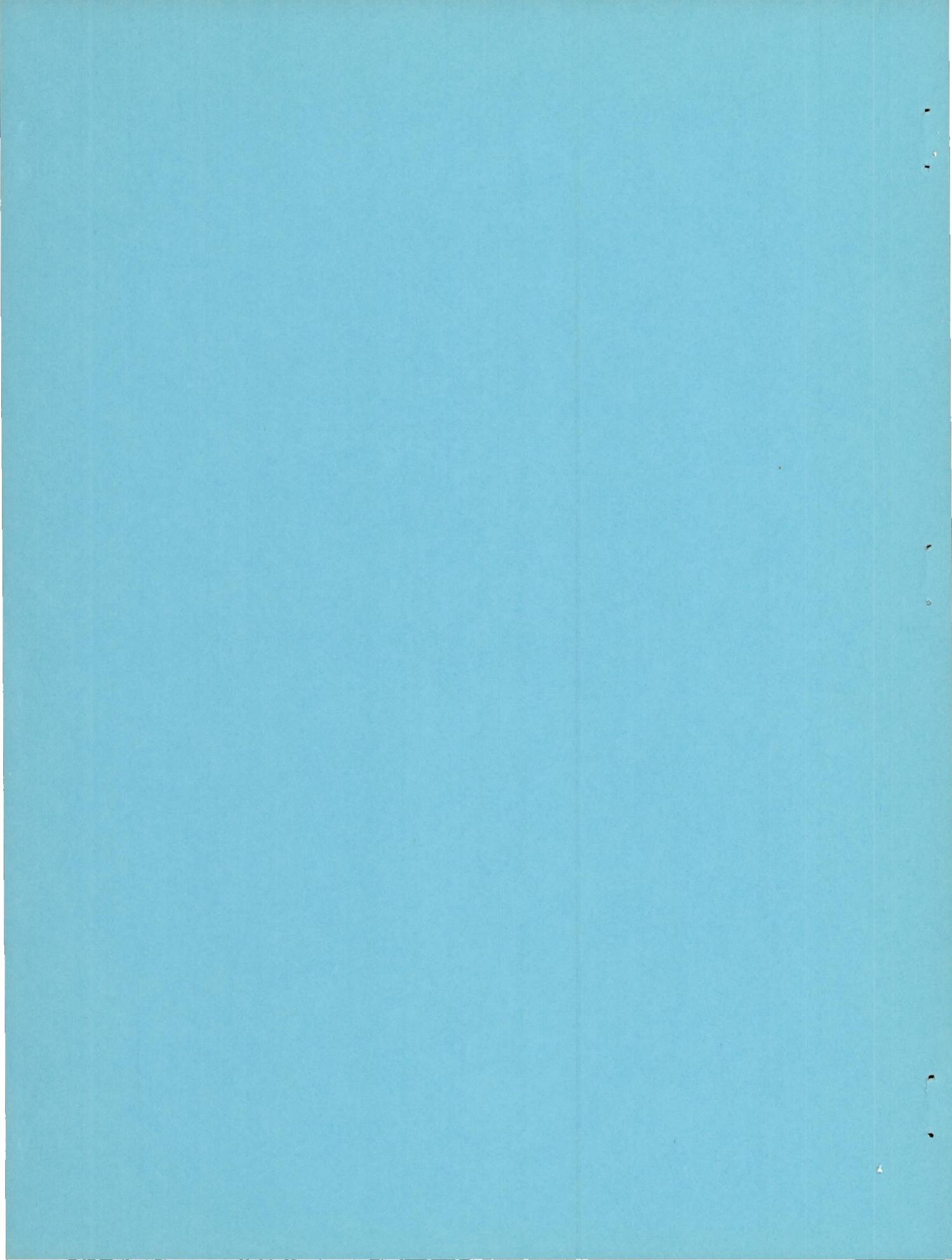
STATIC LATERAL STABILITY CHARACTERISTICS
AT MACH NUMBERS OF 1.40 AND 1.59

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NATIONAL ADVISORY COMMITTEE
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SUMMARY

An investigation has been conducted in the Langley 4- by 4-foot supersonic tunnel to determine the static lateral stability characteristics of a supersonic aircraft configuration at Mach numbers of 1.40 and 1.59. The model had a 40° sweepback wing with 10-percent-thick circular-arc sections normal to the quarter-chord line.

The results of the investigation indicated high directional stability that decreased with increasing Mach number and positive effective dihedral that was essentially invariant with lift coefficient and Mach number.

INTRODUCTION

A comprehensive wind-tunnel investigation has been conducted in the Langley 4- by 4-foot supersonic tunnel to determine the stability and control characteristics as well as the general aerodynamic characteristics of a supersonic aircraft configuration. The model had a wing with 40° sweepback at the quarter-chord line, aspect ratio 4, taper ratio 0.5, and 10-percent-thick circular-arc sections normal to the quarter-chord line.

The longitudinal stability and control characteristics of the model at a Mach number of 1.40 are presented in reference 1. Pressure measurements over the fuselage of the model are presented in reference 2 for

a Mach number of 1.59 and in reference 3 for a Mach number of 1.40. The present paper contains the results of the lateral stability investigation conducted at Mach numbers of 1.40 and 1.59.

COEFFICIENTS AND SYMBOLS

The results of the tests are presented as standard NACA coefficients of forces and moments. The data are referred to the stability-axes system (fig. 1) with the reference center of gravity at 25 percent of the mean aerodynamic chord.

The coefficients and symbols are defined as follows:

c_L	lift coefficient (Lift/ qS where Lift = -Z)
c_X	longitudinal-force coefficient (X/qS)
c_Y	lateral-force coefficient (Y/qS)
c_l	rolling-moment coefficient (L/qSb)
c_m	pitching-moment coefficient ($M'/qS\bar{c}$)
c_n	yawing-moment coefficient (N/qSb)
Z	force along Z-axis, pounds
X	force along X-axis, pounds
Y	force along Y-axis, pounds
L	moment about X-axis, pound-feet
M'	moment about Y-axis, pound-feet
N	moment about Z-axis, pound-feet
q	free-stream dynamic pressure, pounds per square foot
M	Mach number
S	wing area, square feet
b	wing span, feet

\bar{c}	wing mean aerodynamic chord, feet	$\left(\frac{2}{S} \int_0^{b/2} c^2 dy \right)$
c	airfoil-section chord, feet	
y	distance along wing span, feet	
α	angle of attack of fuselage center line, degrees	
α_t	stabilizer incidence angle with respect to fuselage center line, degrees	
ψ	angle of yaw, degrees	
$C_{Y\psi}$	lateral-force parameter, rate of change of lateral-force coefficient with angle of yaw, per degree	$\left(\frac{\partial C_Y}{\partial \psi} \right)$
$C_{l\psi}$	effective-dihedral parameter, rate of change of rolling-moment coefficient with angle of yaw, per degree	$\left(\frac{\partial C_l}{\partial \psi} \right)$
$C_{n\psi}$	directional-stability parameter, rate of change of yawing-moment coefficient with angle of yaw, per degree	$\left(\frac{\partial C_n}{\partial \psi} \right)$
$C_{l\psi C_L}$	rate of change of effective-dihedral parameter with lift coefficient	$\left(\frac{\partial C_l}{\partial C_L} \right) \psi$

MODEL AND APPARATUS

A three-view drawing of the model is shown in figure 2, and the geometric characteristics of the model are given in table I. The model mounted for testing in the tunnel is shown in figure 3.

The model had a wing with 40° sweepback at the quarter-chord line, aspect ratio 4, taper ratio 0.5, and 10-percent-thick circular-arc sections normal to the quarter-chord line. The 20-percent-chord flat-side ailerons having a trailing-edge thickness 0.5 of the hinge-line thickness were installed on the outboard 50 percent of the wing semi-spans. The wing was at a 3° incidence angle with respect to the fuselage center line.

The model was mounted on a sting support and its angle in the horizontal plane was remotely controlled in such a manner that the model remained essentially in the center of the test section. With the model mounted so that the wings were vertical, tests could be made through an angle-of-attack range (see fig. 3(a)). With the model rotated 90° (wings horizontal), the angle-of-attack mechanism was used to provide angles of yaw. (See fig. 3(b).) A straight sting was used for pitch tests at zero yaw and yaw tests at zero angle of attack while stings having 3° and 6° bends were used for pitch tests at 3° and 6° yaw and for yaw tests at 3° and 6° angle of attack.

The stabilizer angle could be remotely controlled by means of an electric motor located within the fuselage of the model.

Forces and moments on the model were measured by means of an internal six-component strain-gage balance. Some details of the balance and support system are included in reference 1.

The tests were conducted in the Langley 4- by 4-foot supersonic tunnel which is described in reference 2.

TESTS

Test Conditions

The test conditions are summarized in the following table:

Mach number	Stagnation pressure (atm)	Stagnation temperature (°F)	Dew point (°F)	Dyanmic pressure (lb/sq ft)	Reynolds number (based on \bar{c})
1.40	0.25	110	-30	229	600,000
1.59	.25	110	-35	223	575,000

Calibration data for the Mach number 1.40 nozzle are presented in reference 3 and for the Mach number 1.59 nozzle in reference 2.

Corrections and Accuracy

No corrections due to sting interference were applied to the data. Though it is believed that the sting effects are small, the exact magnitude is not known. Some repeat runs made with various bent stings showed

excellent agreement and indicated that whatever sting effects exist are independent of whether the sting is bent or straight. Base-pressure measurements at a Mach number of 1.59 indicated a drag correction that was within the accuracy of the scale readings for the low angles of attack. For the angle-of-attack range from 4° to 10° , the correction would result in a drag reduction of about 1 percent. Since the maximum sting deflection under load was within the accuracy of the angle measurements, no angle-of-attack or yaw correction was required.

The maximum uncertainties in the aerodynamic coefficients due to the balance system are as follows:

The accuracy of the angle of attack was about $\pm 0.05^\circ$, the tail incidence about $\pm 0.10^\circ$, and the dynamic pressure about 0.25 percent.

The variation in Mach number in the vicinity of the model due to flow irregularities is about ± 0.01 . At a Mach number of 1.40 (reference 3), the flow angularity in the horizontal plane is about $\pm 0.2^\circ$ and in the vertical plane, about 0.27° to -0.11° . At a Mach number of 1.59 (reference 2), the flow angularity in the horizontal plane is about 0° to 0.20° and in the vertical plane about 0.30° to 0° . Tests made with the model in the horizontal and in the vertical positions but at the same attitude showed excellent agreement indicating the effect of stream irregularity to be negligible.

Test Procedure

Tests were made through a yaw range up to 10° at angles of attack of 0° and 6° at $M = 1.40$ and at angles of attack of 0° , 3° , and 6° at $M = 1.59$. Tests were made through an angle-of-attack range up to 10° at angles of yaw of 0° and 6° at $M = 1.40$ and at angles of yaw of 0° , 3° , and 6° at $M = 1.59$.

Tests with the horizontal and vertical tails removed were made through the angle-of-yaw range at 0° angle of attack at $M = 1.40$ and at 0° and 3° angle of attack at $M = 1.59$, and through an angle-of-attack range at 0° and 3° angle of yaw at $M = 1.59$.

RESULTS AND DISCUSSION

The variation of the aerodynamic characteristics with angle of yaw for the complete model and for the model with the tail off is presented in figures 4 and 5 for Mach numbers of 1.40 and 1.59, respectively. In general, the variations of lateral-force coefficient, yawing-moment coefficient, and rolling-moment coefficient with angle of yaw are quite linear and vary only slightly with angle of attack. There is little change in lift coefficient with angle of yaw and the longitudinal-force coefficient remains essentially constant since, in the stability-axes system, the X-axis yaws with the model. The drag force parallel to the relative wind can be obtained by combining components of the lateral- and longitudinal-force coefficients in the stream direction. The pitching-moment coefficient varies slightly with angle of yaw but the results of longitudinal tests (reference 1 for $M = 1.40$ and unpublished results for $M = 1.59$) indicate that longitudinal trim could be easily maintained.

The variation of the lateral-stability parameters with Mach number at zero angle of attack is presented in figure 6 together with the low-speed values obtained from reference 4. The lateral-force parameter $C_{Y\psi}$ at $M = 1.40$ is approximately the same as that obtained at low speed for both the complete model and the tail-off configuration. Since the tail contribution to the lateral-force parameter $\Delta C_{Y\psi}$ is about the same, apparently the vertical-tail lift-curve slope at $M = 1.40$ is about the same as the low-speed value. At $M = 1.59$, $C_{Y\psi}$ is somewhat less for the complete model but about the same for the tail-off configuration, which probably indicates a decrease in the vertical-tail lift-curve slope with increasing Mach number.

The directional stability $C_{n\psi}$ for the complete model is considerably greater than that obtained at subsonic speeds. With the tail removed, however, the directional stability is about the same as that obtained at subsonic speeds. Inasmuch as $\Delta C_{n\psi}$ for $M = 1.40$ corresponds to the low-speed value, the increase in directional stability probably results from a rearward shift of the center of pressure of the lateral forces produced by the tail. The directional stability at $M = 1.59$ is less than at $M = 1.40$, the decrease being directly proportional to the decrease in $\Delta C_{n\psi}$.

The rolling moment due to yaw or effective-dihedral parameter $C_{l\psi}$ indicates a positive value for the complete model that is about the same for both Mach numbers. Unlike the subsonic case, all of the positive effective dihedral is contributed by the vertical tail as shown by the negative value of $C_{l\psi}$ with the tail removed. This negative $C_{l\psi}$ might

be attributed to the effective change in wing sweep as the model is yawed which, in this Mach number range, might result in a decrease in lift of the advancing wing and an increase in lift of the receding wing - an effect opposite to that experienced at low speeds. Interference effects between the fuselage pressure field and the upper surface of the wing might also contribute to the negative effective dihedral in the same manner as at low speeds. Inasmuch as the vertical tail contributes all of the positive effective dihedral, it is important to know the effects of rudder deflection on $C_{l\psi}$. Tests made to determine the directional control characteristics (unpublished results) indicate positive effective dihedral with controls fixed. However, the variation of C_l with ψ for zero yawing moment ($C_n = 0$) indicates a dihedral effect that is slightly negative at $M = 1.40$ and slightly positive at $M = 1.59$.

The increment of $C_{l\psi}$ resulting from the addition of the tail is greater at $M = 1.40$ than at low speeds. This probably results from a shift of the vertical-tail center of pressure toward the tip of the vertical tail. The tail contribution is less at $M = 1.59$ by an amount proportional to the decrease in $\Delta C_{Y\psi}$, but little change occurs in $C_{l\psi}$ for the complete model because of an increase in effective dihedral of the wing-fuselage combination. The effective dihedral of the wing-fuselage combination is higher at $M = 1.59$ than at $M = 1.40$ because of the decrease in the rate of change of lift with Mach number and possibly because of a reduction in fuselage-wing interference effects.

The variation of the lateral characteristics through the lift-coefficient range for various angles of yaw is shown in figures 7 and 8 for Mach numbers of 1.40 and 1.59, respectively. These data were obtained using various stabilizer deflections so that the model remained trimmed in pitch since some data obtained at $M = 1.59$ for an angle of attack of 4° and an angle of yaw of 6° indicated slight decreases in C_y , C_n , and C_l as the stabilizer incidence was changed from 4° to -10° . This effect is probably a result of interference between the stabilizer and vertical tail that would vary as the lift of the stabilizer varied. The increment of rolling moment contributed by the stabilizer would also vary with the lift of the stabilizer. These effects of stabilizer incidence on the lateral characteristics, although small, were measurable and may assume greater importance for other configurations. Included in figures 7 and 8 for comparison are values (large symbols) taken from the yaw tests (figs. 4 and 5) wherein the model was mounted with the wings in a horizontal plane. The conformity of the data is an indication of the small effect of changing the sting and of the tunnel flow angularity on the test results.

The variation of the lateral-stability parameters throughout the lift-coefficient range as obtained by cross-plotting from figures 7 and 8 is presented in figure 9. The symbols appearing in figure 9 represent values measured from the yaw tests (figs. 4 and 5) and are included for comparison. The lateral-stability parameters for both Mach

numbers vary only slightly through the trim-lift-coefficient range which extends from about $C_L \approx 0$ to $C_L \approx 0.37$. (The lift curves for both Mach numbers are given in figure 10.) Tail-off characteristics through the lift range were obtained only at $M = 1.59$.

For the complete model, the slight decrease in $C_{Y\psi}$ and $C_{n\psi}$ with increasing lift coefficient (fig. 9) may result partly from a blanketing effect of the wing and fuselage on the vertical tail and partly from interference between the stabilizer and vertical tail. There is little change in $C_{Y\psi}$ and $C_{n\psi}$ with lift coefficient for the model with the tail off.

The slight variation of $C_{l\psi}$ with lift coefficient for both the complete model and the tail-off configuration is in contrast to the increase usually obtained at low speeds for similar configurations (reference 4, for example). This difference is a result of various effects that cannot be completely isolated. For the model with the tail off, a negative value of $C_{l\psi}$ occurs at $C_L = 0$ although the wing has positive geometric

dihedral. As already pointed out, this may be due in part to an interference effect between the fuselage and wing and to the effective change in wing sweep as the model is yawed. If the effect of wing sweep is such that the advancing wing has the lower lift-curve slope, it would be expected that the rate of change of effective dihedral with lift coefficient $C_{l\psi C_L}$ would be negative. However, a slightly positive value

of $C_{l\psi C_L}$ is indicated by the tail-off data for $M = 1.59$. This variation might be influenced by the fuselage itself which should provide a positive increment of $C_{l\psi C_L}$. The effect of positive geometric dihedral

should also result in a positive increment of $C_{l\psi C_L}$. In any case, the

slightly positive value of $C_{l\psi C_L}$ for the model with the tail removed

indicates that, in this Mach number range, the increment in $C_{l\psi C_L}$ due to the wing alone is small compared with that obtained at low speeds. Installation of the vertical tail provides a positive increment of $C_{l\psi}$ and a negative increment of $C_{l\psi C_L}$ in the same manner as at low speeds and

the resultant $C_{l\psi C_L}$ for the complete model is very low. The slightly

higher value of $C_{l\psi C_L}$ for the complete model at $M = 1.40$ indicates

that $C_{l\psi C_L}$ for the tail-off case is probably greater at $M = 1.40$ than

at $M = 1.59$ inasmuch as the negative value of $C_{l\Psi}C_L$ resulting from the vertical tail should be greater at $M = 1.40$.

A comparison of $C_{Y\Psi}$ and $C_{n\Psi}$ at $C_L \approx 0$ with results of tests of a similar configuration in the Langley 9-inch supersonic tunnel (reference 5) is given in figure 11. The Reynolds number for the tests in the Langley 9-inch supersonic tunnel varies from 410,000 at $M = 1.55$ to 310,000 at $M = 2.32$. Results of the present tests indicate a slightly lower value of $C_{Y\Psi}$ and a proportionately lower value of $C_{n\Psi}$. Some of the difference is a consequence of a small opening made in the vertical tail of the present model to permit deflection of the horizontal tail. Tests made with the opening sealed indicated that $C_{Y\Psi}$ and $C_{n\Psi}$ might be increased about 10 percent. Other factors that might affect the comparison of results are differences in the model mounting, in the balance system, and in the corrections applied to the data of reference 5.

The variation of $C_{n\Psi}$ with Mach number indicates a trend toward neutral directional stability that probably results in part from a decrease in the lift-curve slope of the vertical tail with increasing Mach number.

CONCLUDING REMARKS

Results of the static-lateral-stability investigation conducted at Mach numbers of 1.40 and 1.59 on a model of a supersonic aircraft configuration indicated satisfactory lateral and directional stability characteristics. The model exhibited high directional stability that decreased with increasing Mach number, and positive effective dihedral that was essentially invariant with increasing lift coefficient and Mach number.

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3. Hasel, Lowell E., and Sinclair, Archibald R.: A Pressure-Distribution Investigation of a Supersonic-Aircraft Fuselage and Calibration of the Mach Number 1.40 Nozzle of the Langley 4- by 4-Foot Supersonic Tunnel. NACA RM L50B14a, 1950.
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TABLE I.- GEOMETRIC CHARACTERISTICS OF MODEL

Wing:

Area, sq ft	1.158
Aspect ratio	4
Sweepback of quarter-chord line, deg	40
Taper ratio	0.5
Mean aerodynamic chord	0.557
Airfoil section normal to quarter-chord line	10-percent-thick, circular-arc
Twist, deg	0

Horizontal tail:

Area, sq ft	0.196
Aspect ratio	3.72
Sweepback of quarter-chord line, deg	40
Taper ratio	0.5
Airfoil section	NACA 65-008

Vertical tail:

Area (exposed), sq ft	0.172
Aspect ratio (based on exposed area and span)	1.17
Sweepback of leading edge, deg	40.6
Taper ratio	0.337
Airfoil section, root	NACA 27-010
Airfoil section, tip	NACA 27-008

Fuselage:

Fineness ratio (neglecting canopies)	9.4
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Miscellaneous:

Tail length from $\bar{c}/4$ wing to $\bar{c}_t/4$ tail, ft	0.917
Tail height, wing semispans above fuselage center line	0.153



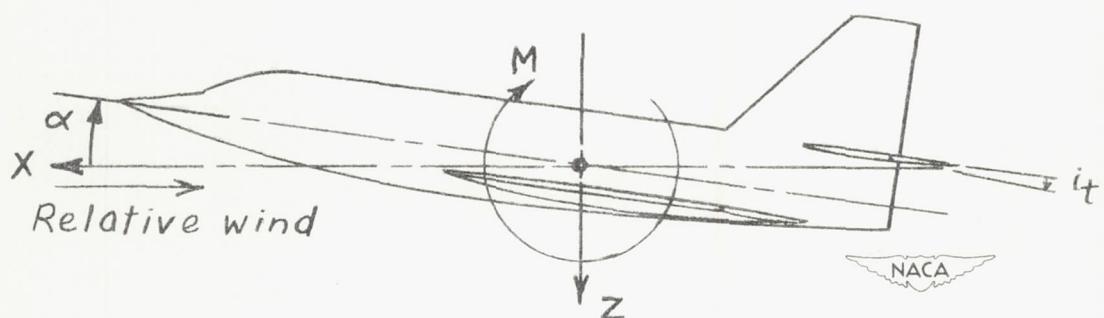
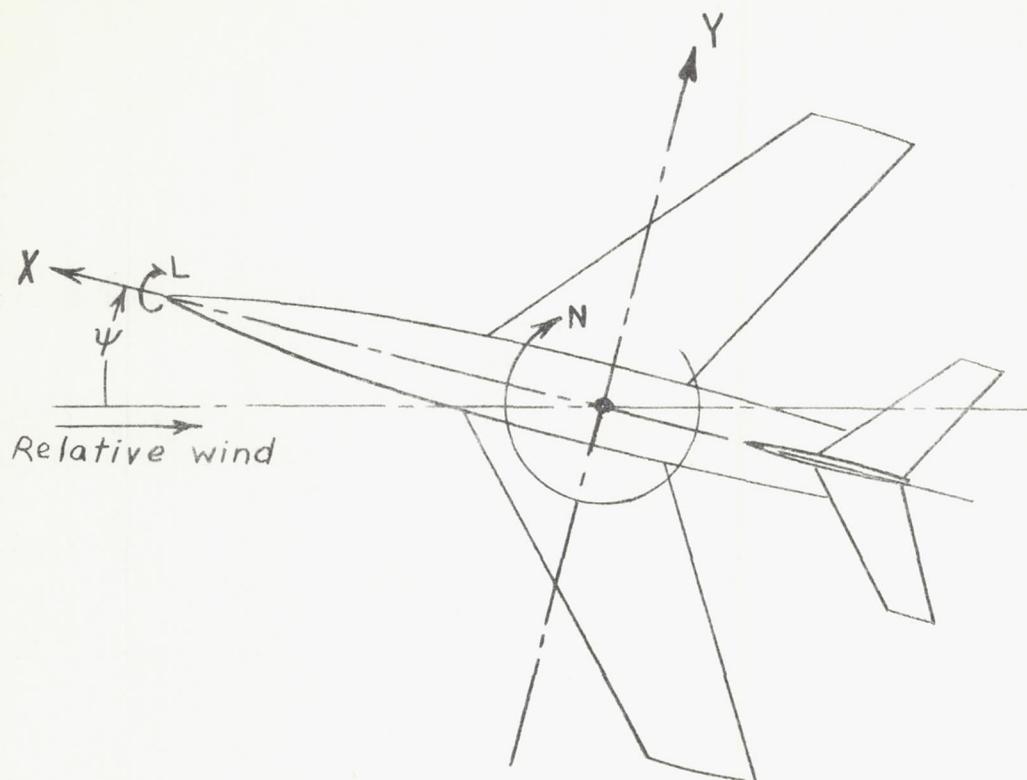


Figure 1.- System of stability axes. Arrows indicate positive values.

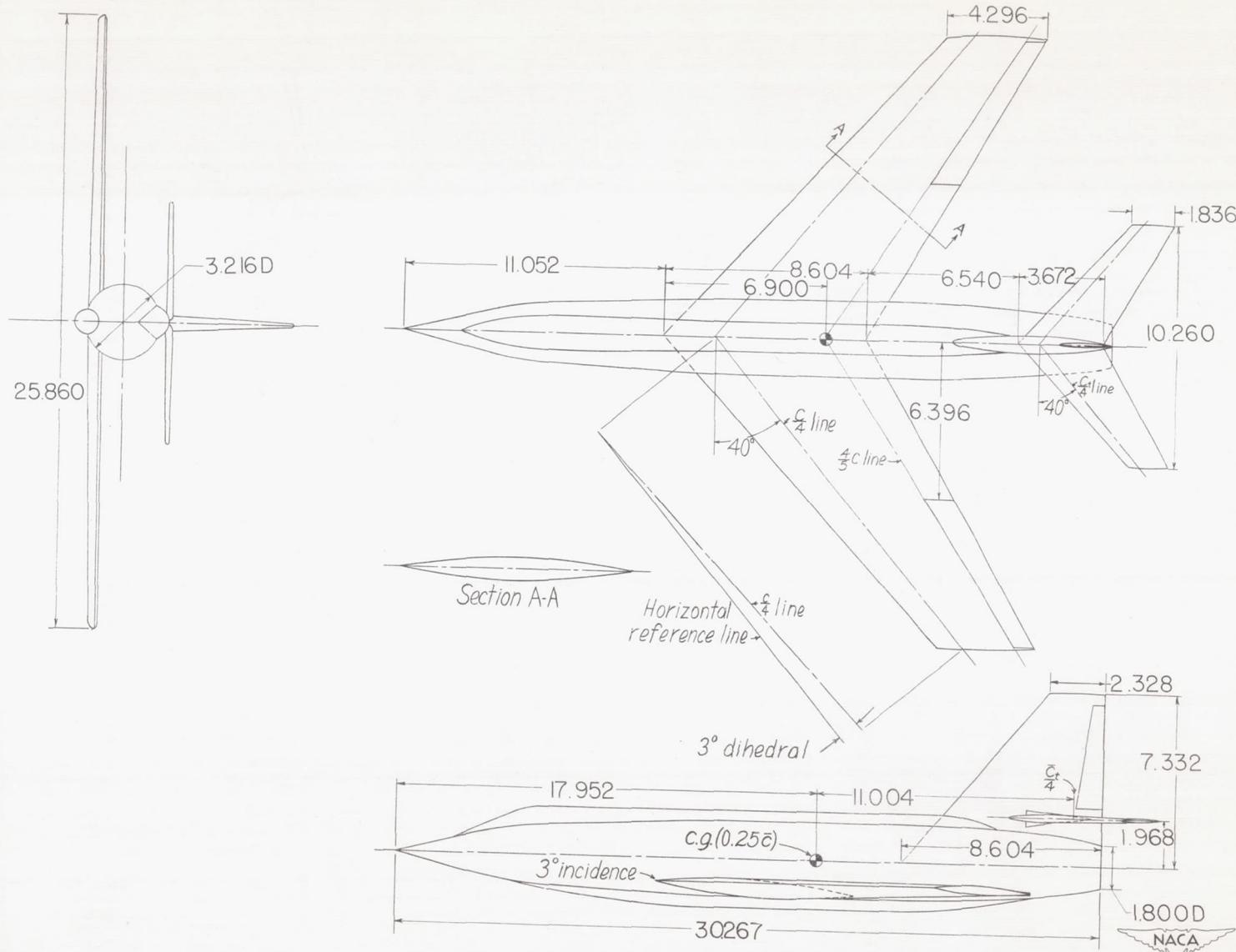
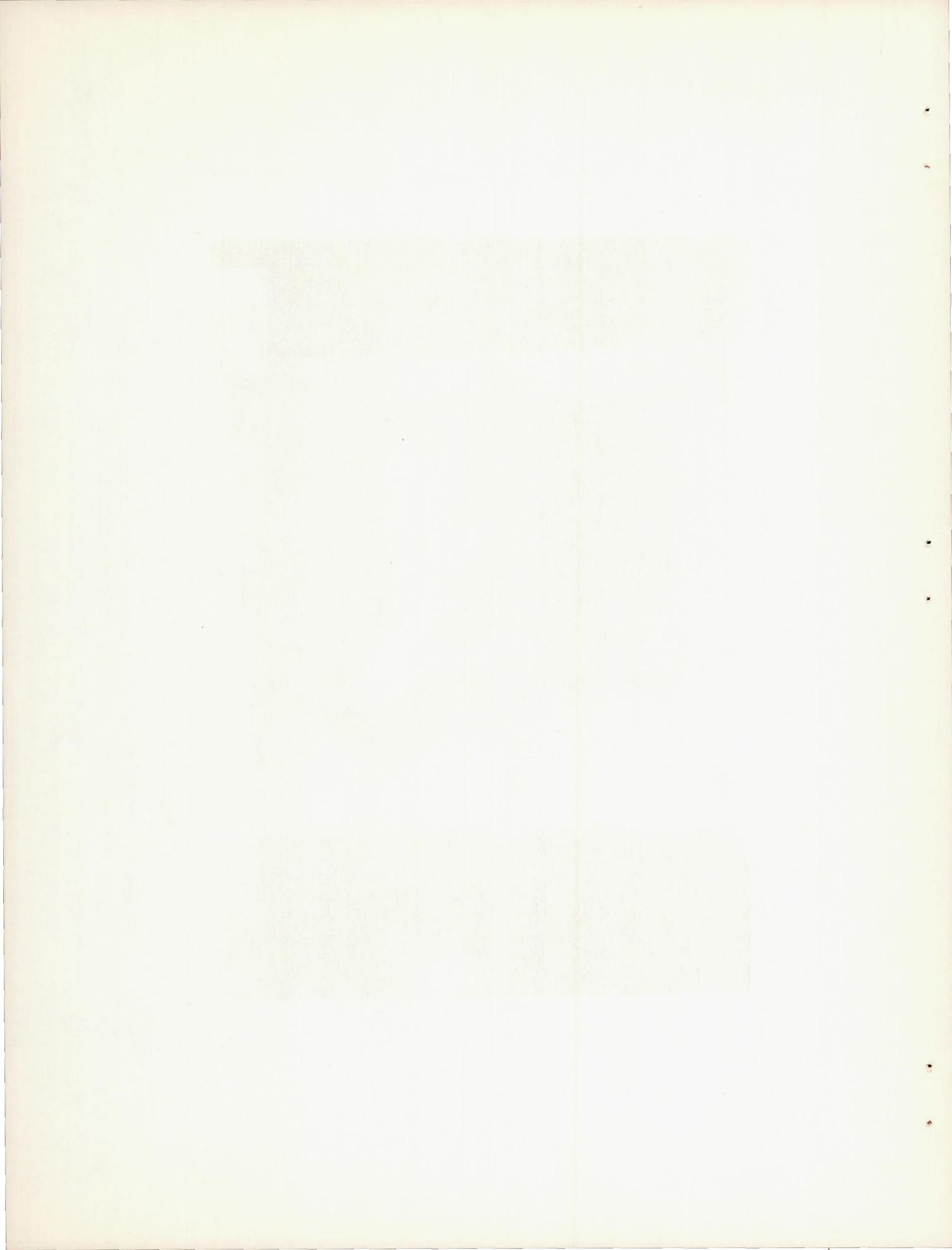
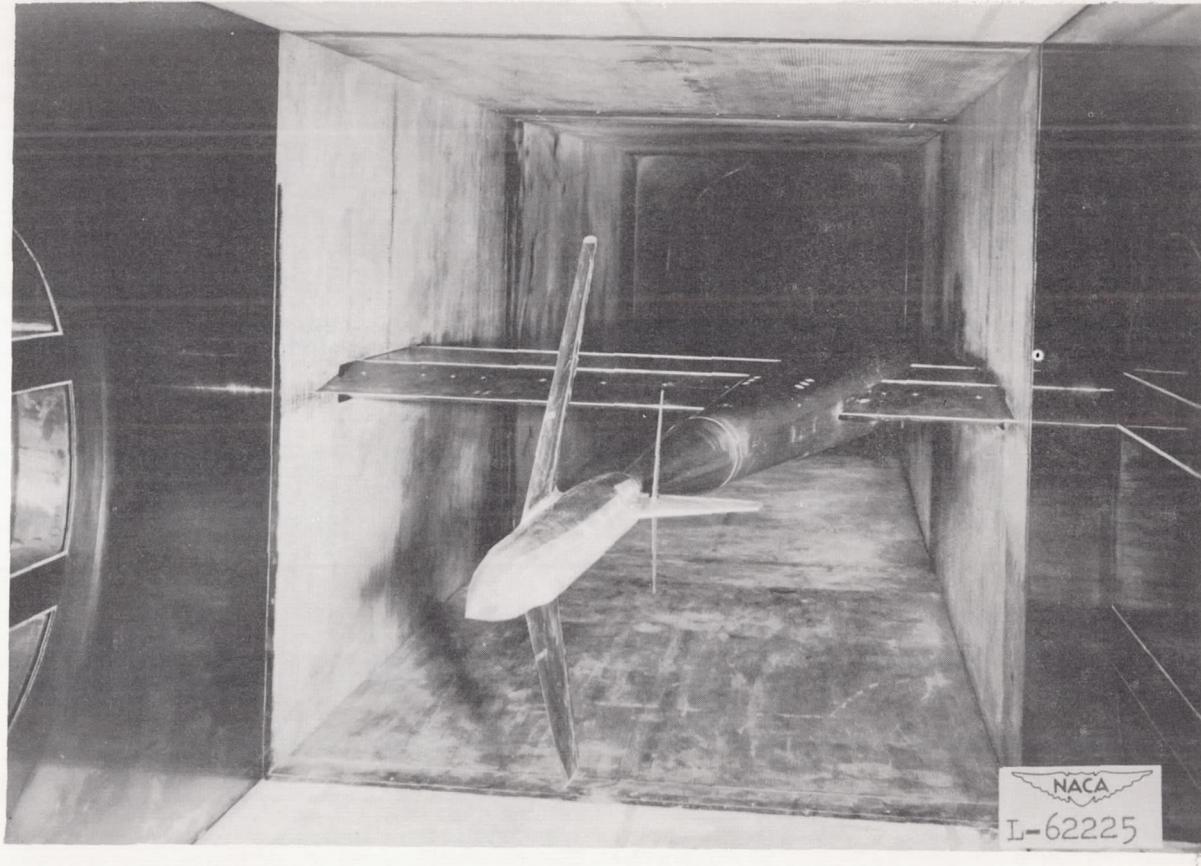


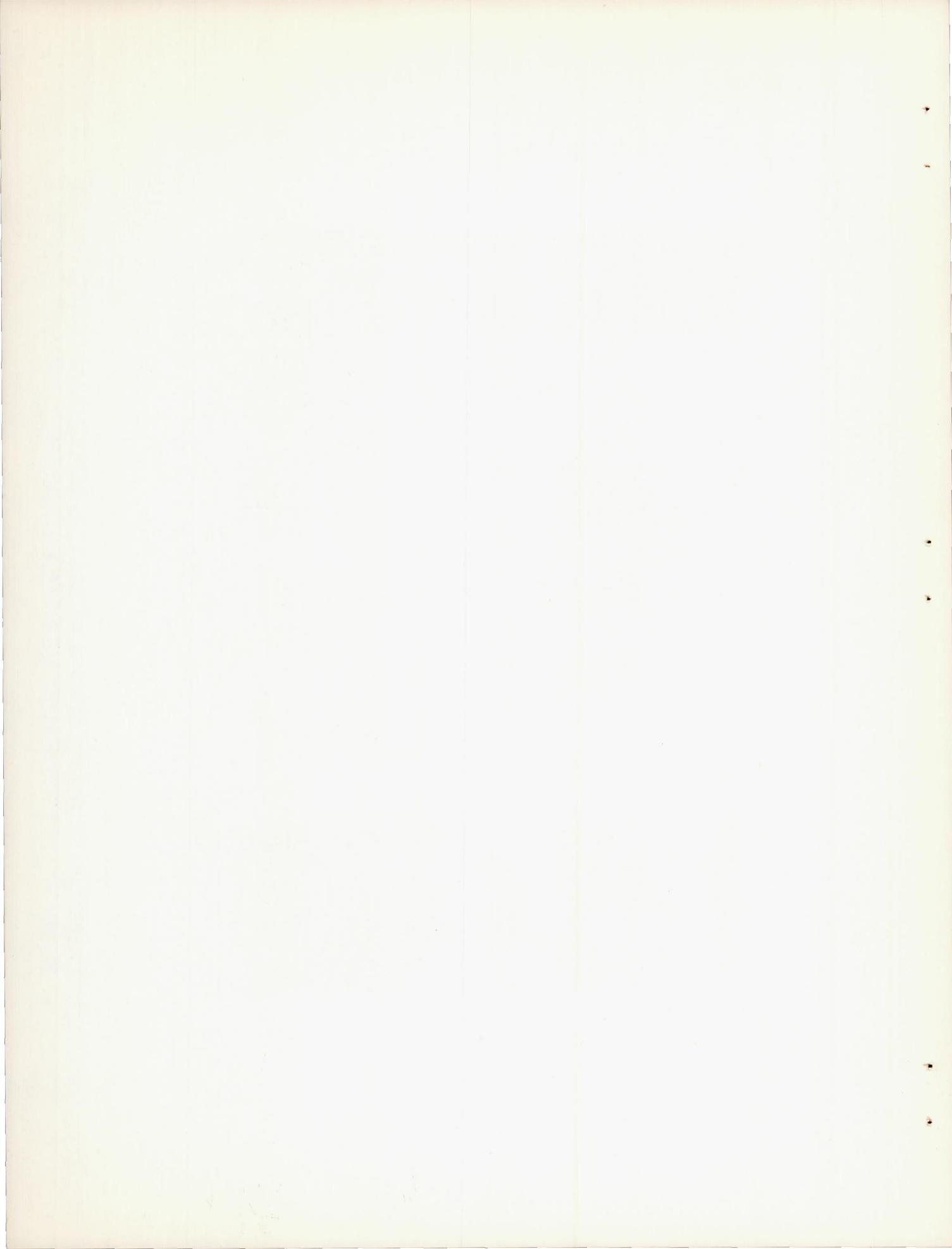
Figure 2.- Details of model of supersonic aircraft configuration.
Dimensions in inches unless otherwise noted.

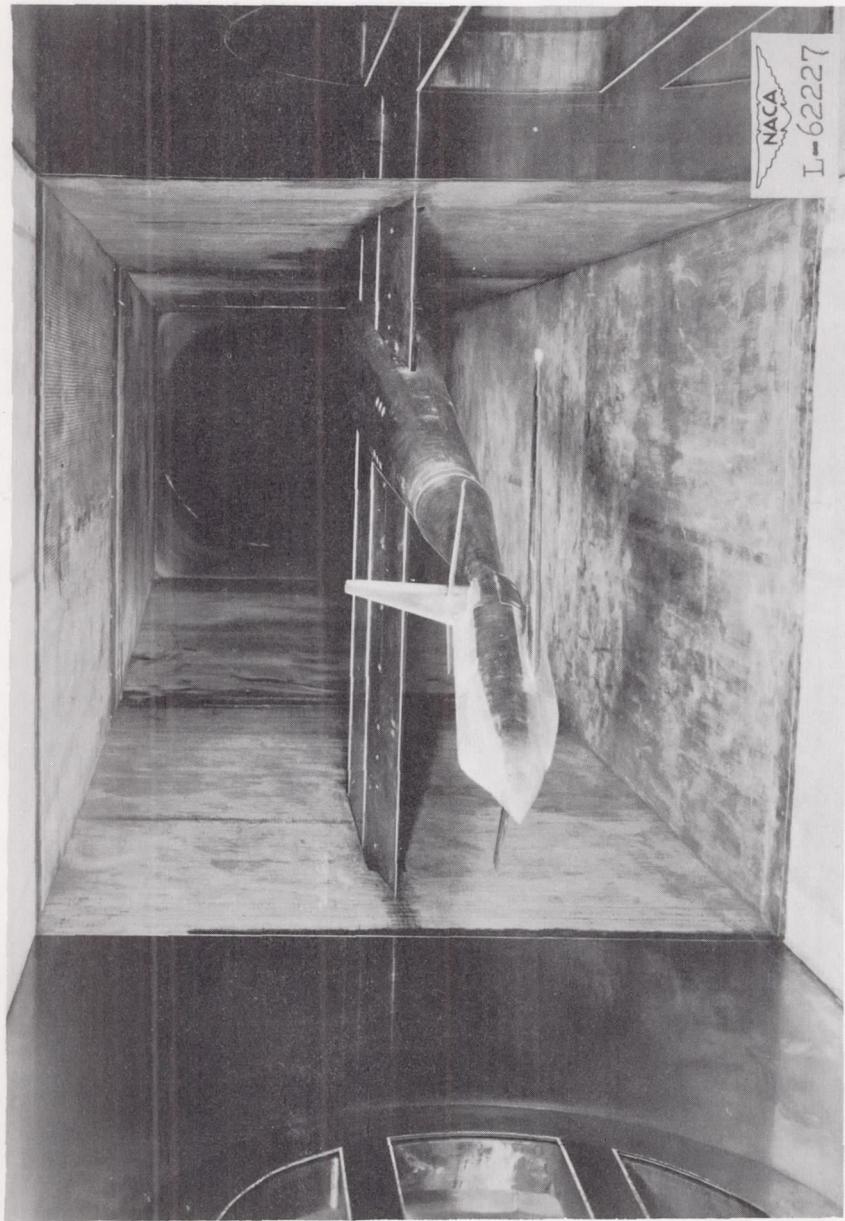




(a) Mounted for pitch tests. $\alpha = -10^\circ$; $\psi = 0^\circ$.

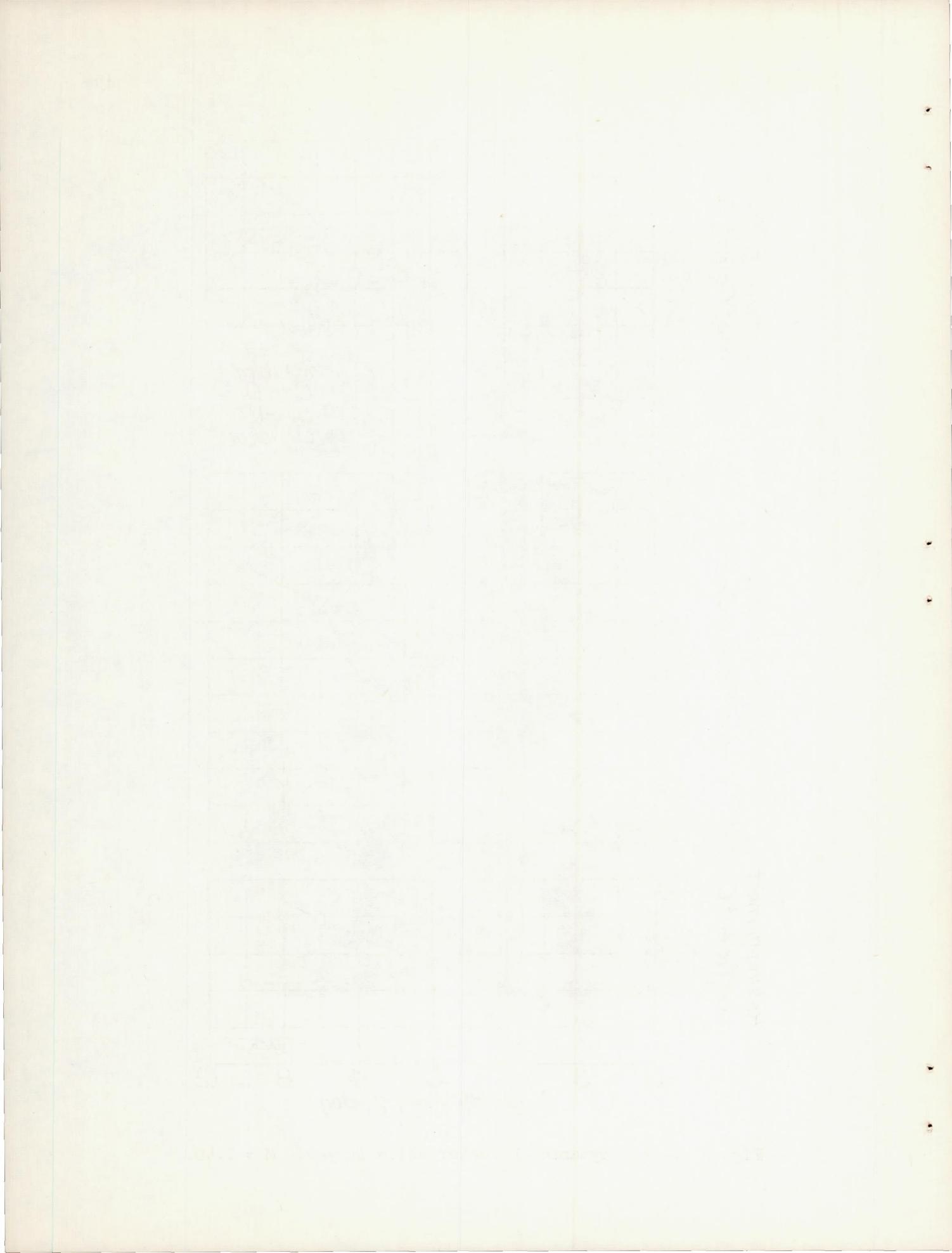
Figure 3.- Complete model of supersonic aircraft mounted in the Langley 4- by 4-foot supersonic tunnel.

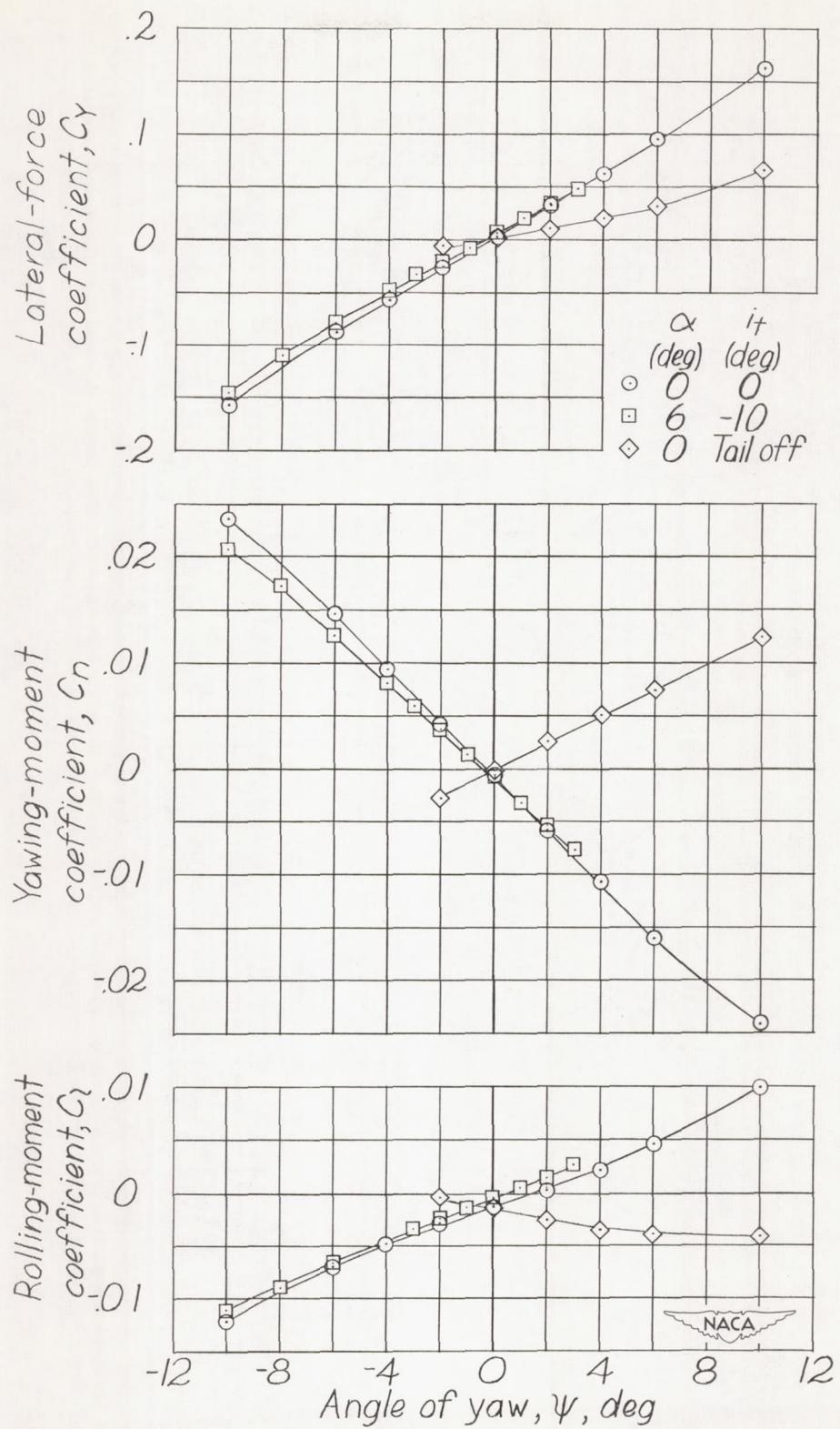




(b) Mounted for yaw tests. $\alpha = 60^\circ$; $\psi = 10^\circ$.

Figure 3.- Concluded.



Figure 4.- Aerodynamic characteristics in yaw. $M = 1.40$.

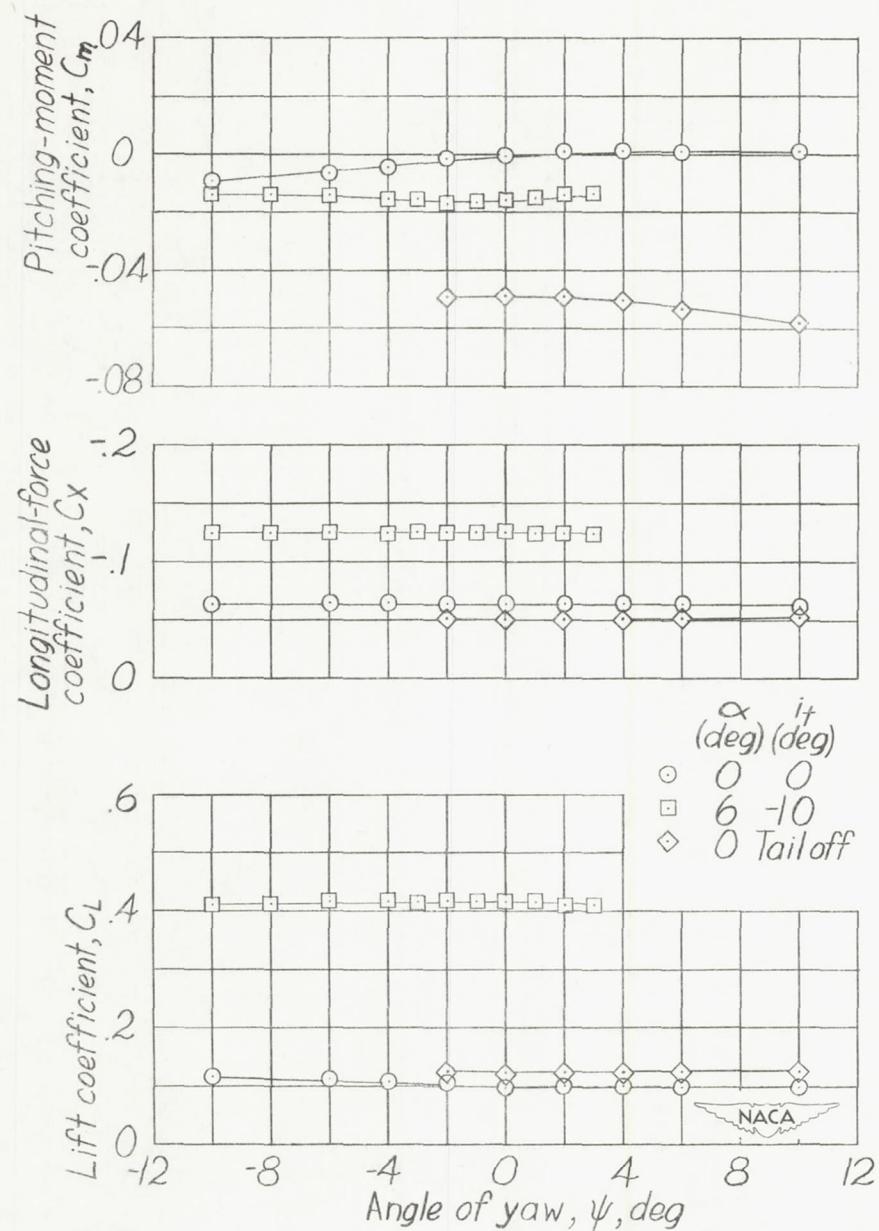
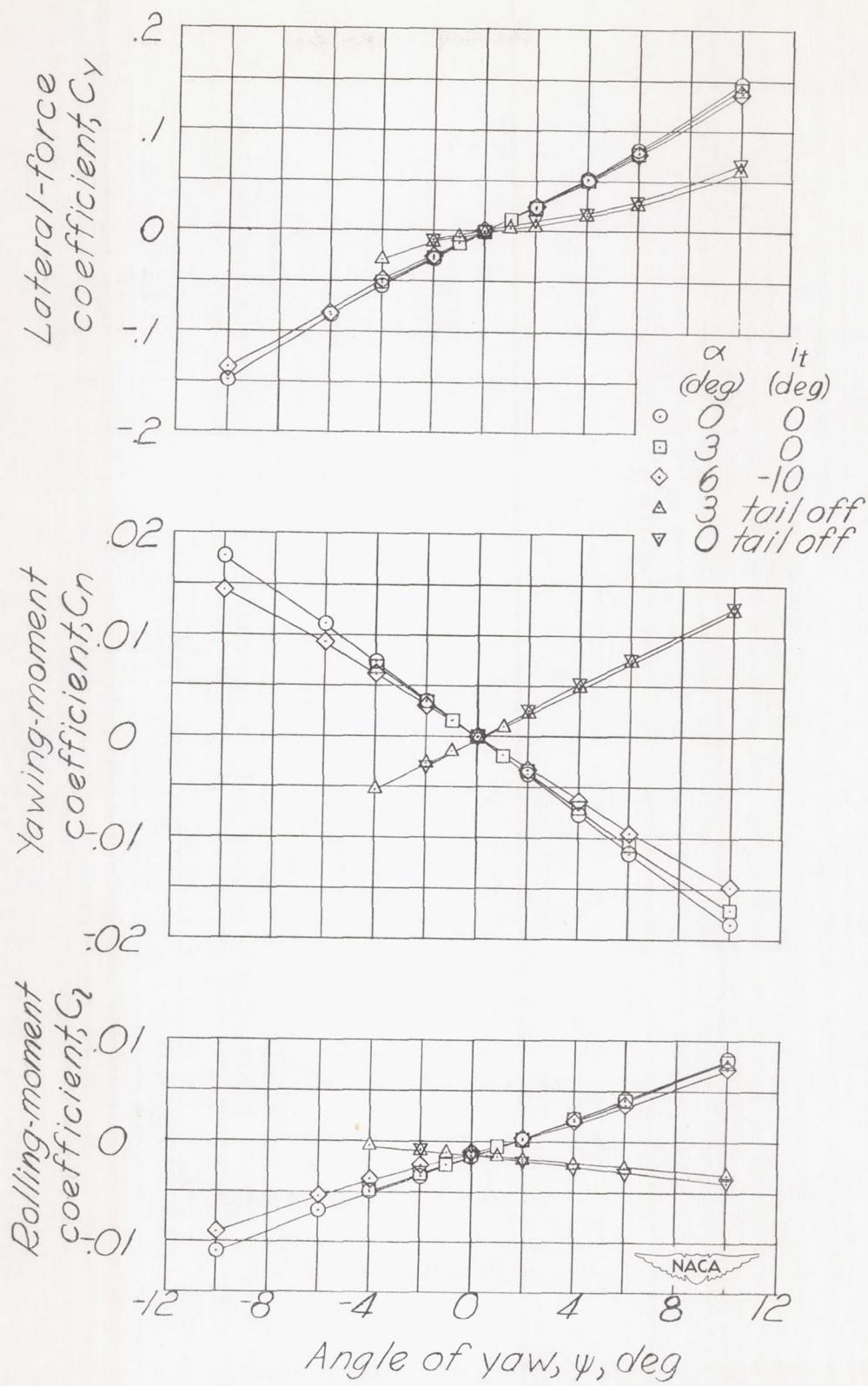


Figure 4.- Concluded.

Figure 5.- Aerodynamic characteristics in yaw. $M = 1.59$.

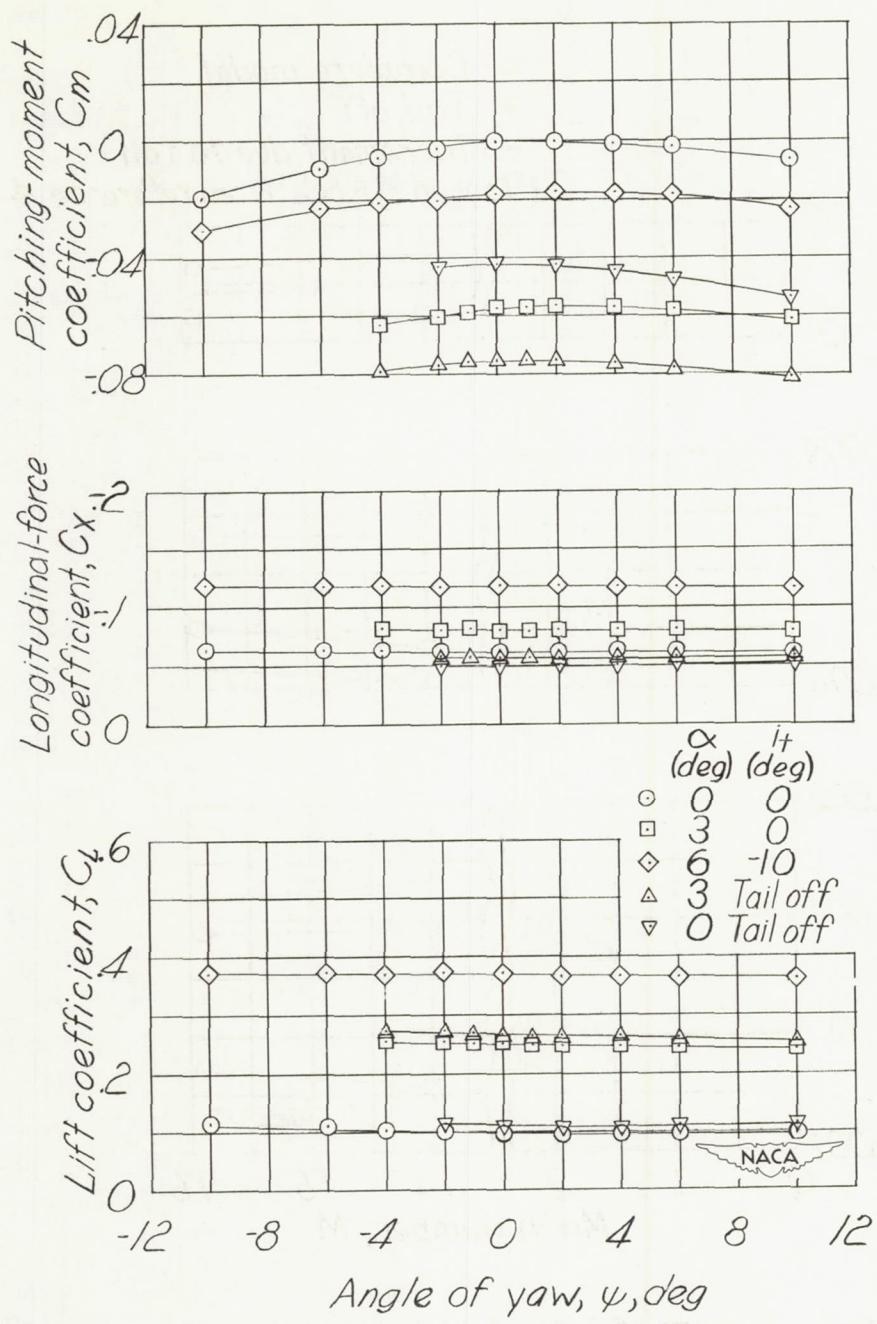
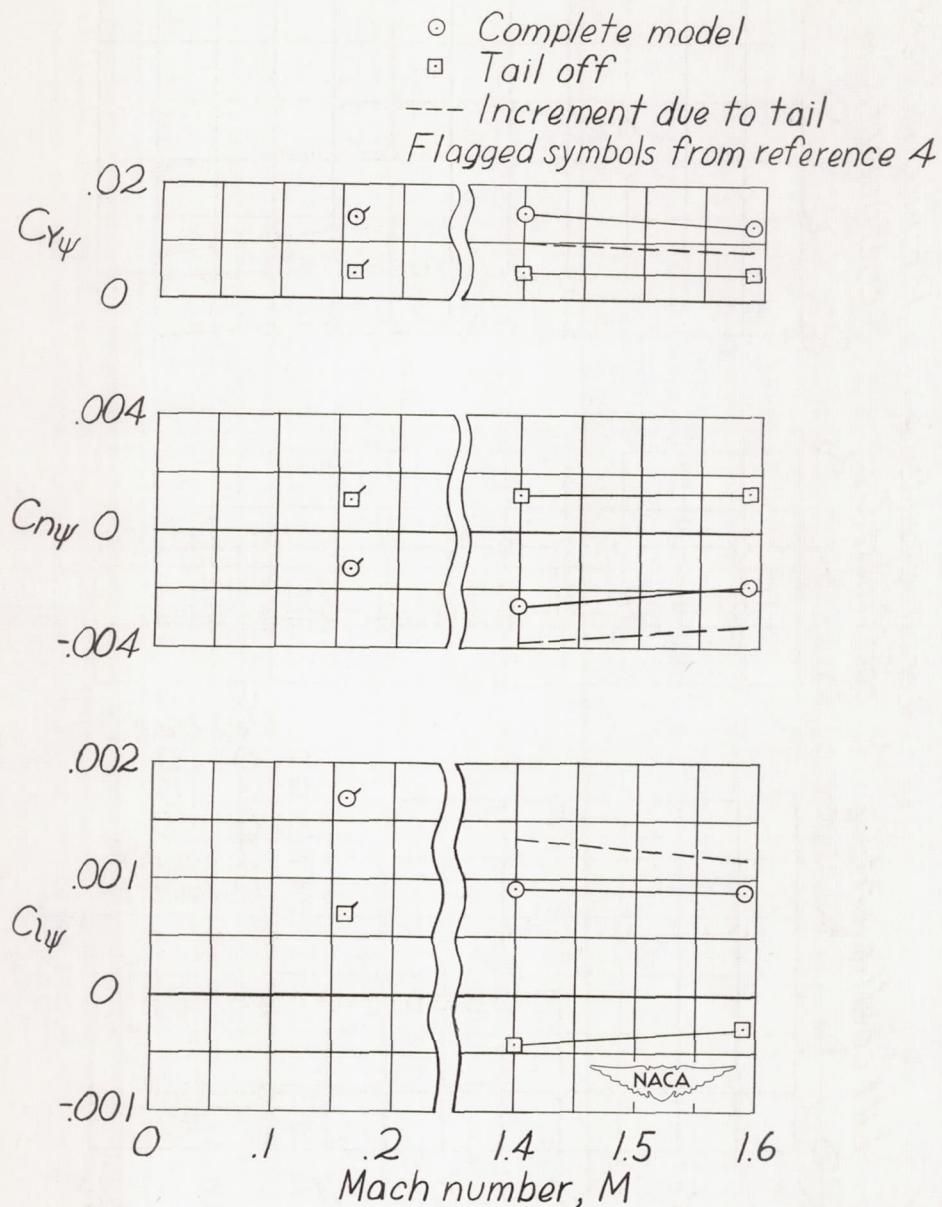


Figure 5.- Concluded.

Figure 6.- Summary of lateral-stability parameters. $\alpha = 0^\circ$.

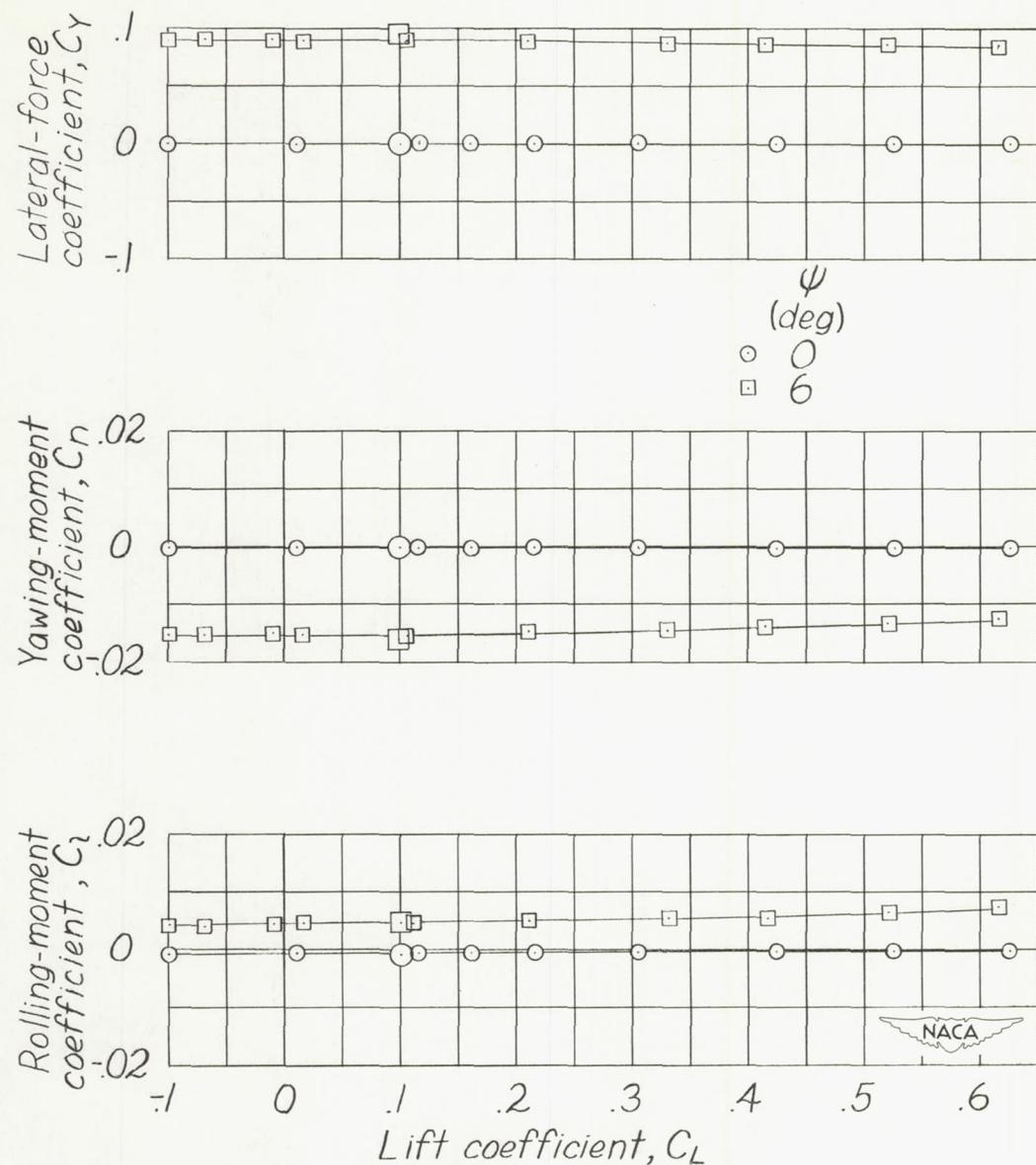


Figure 7.- Effect of yaw on the lateral aerodynamic characteristics in pitch. $M = 1.40$. Large symbols are values from yaw tests.

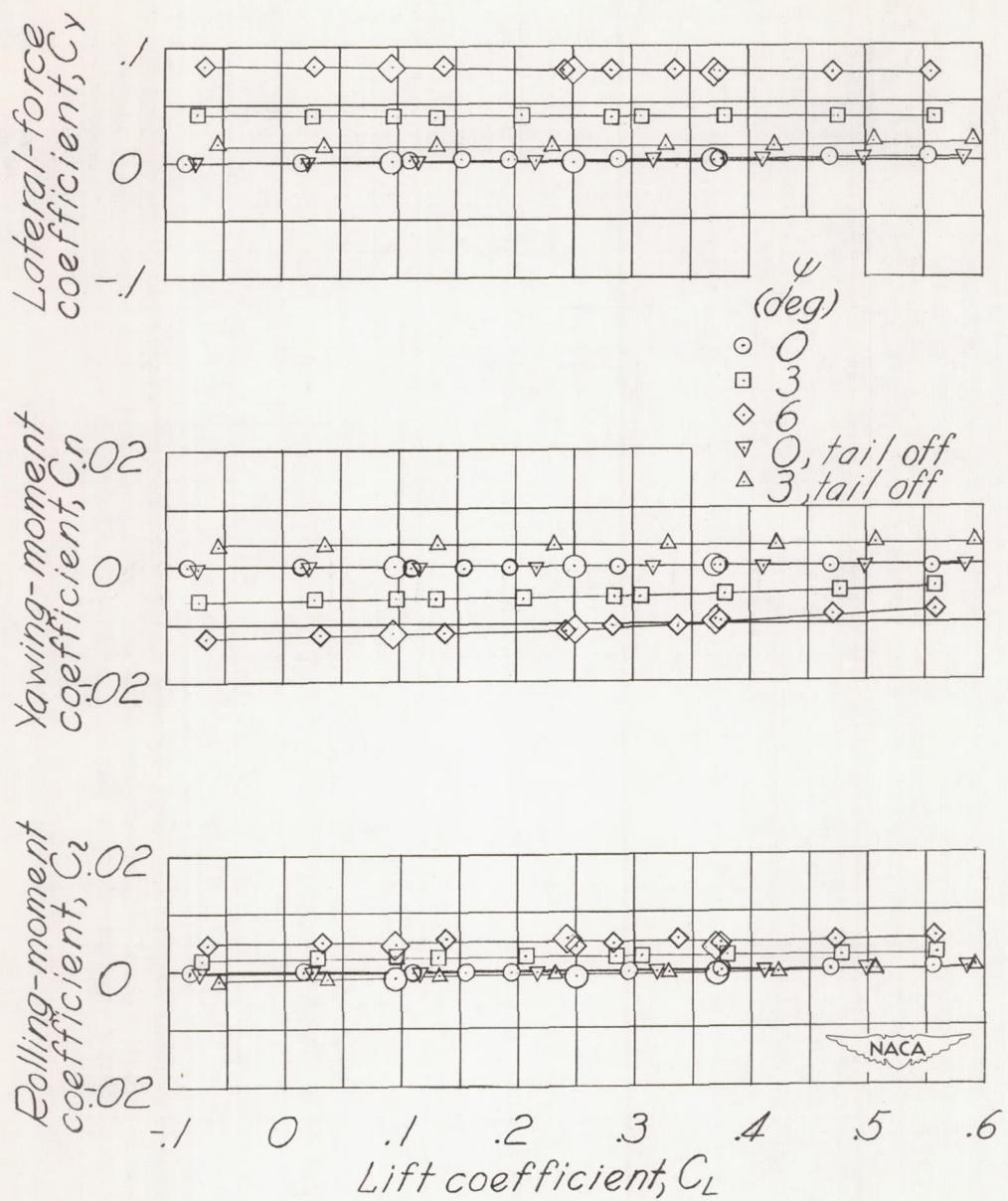


Figure 8.- Effect of yaw on the lateral aerodynamic characteristics in pitch. $M = 1.59$. Large symbols are values from yaw tests.

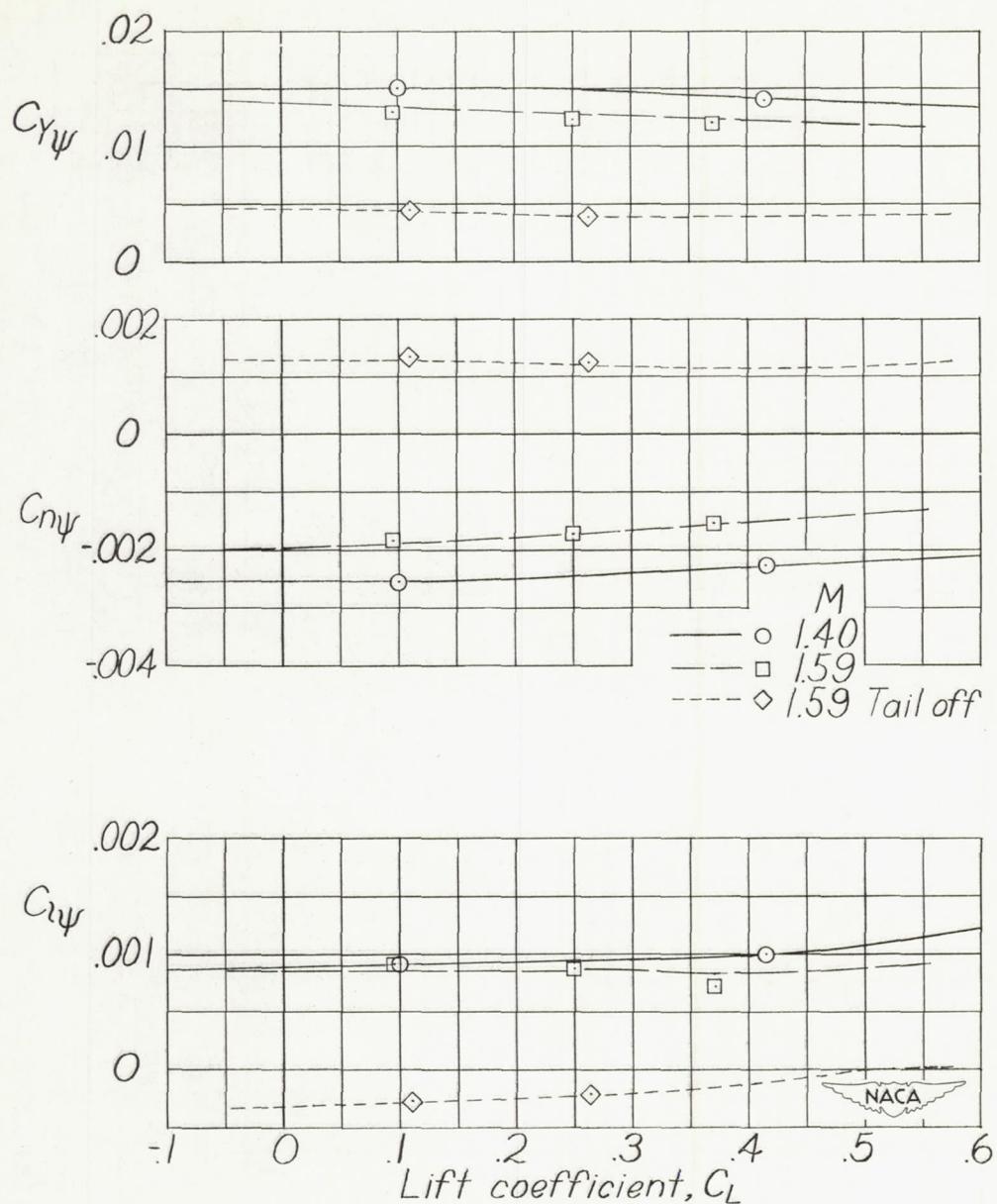


Figure 9.- Variation of the lateral-stability parameters with lift coefficient for Mach numbers of 1.40 and 1.59. Symbols are values from yaw tests.

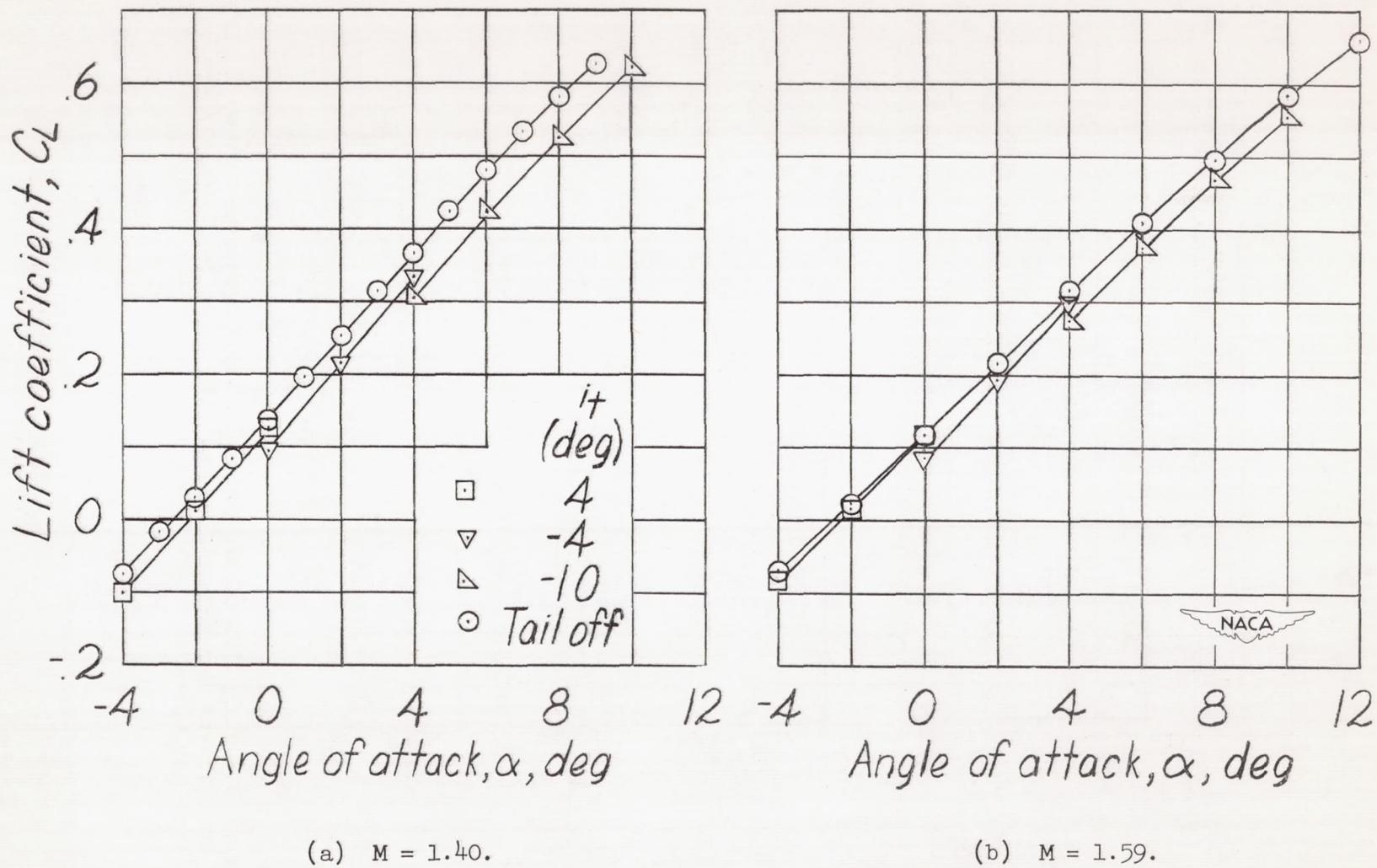
(a) $M = 1.40$.(b) $M = 1.59$.

Figure 10.- Variation of lift coefficient with angle of attack. $\psi = 0^\circ$.

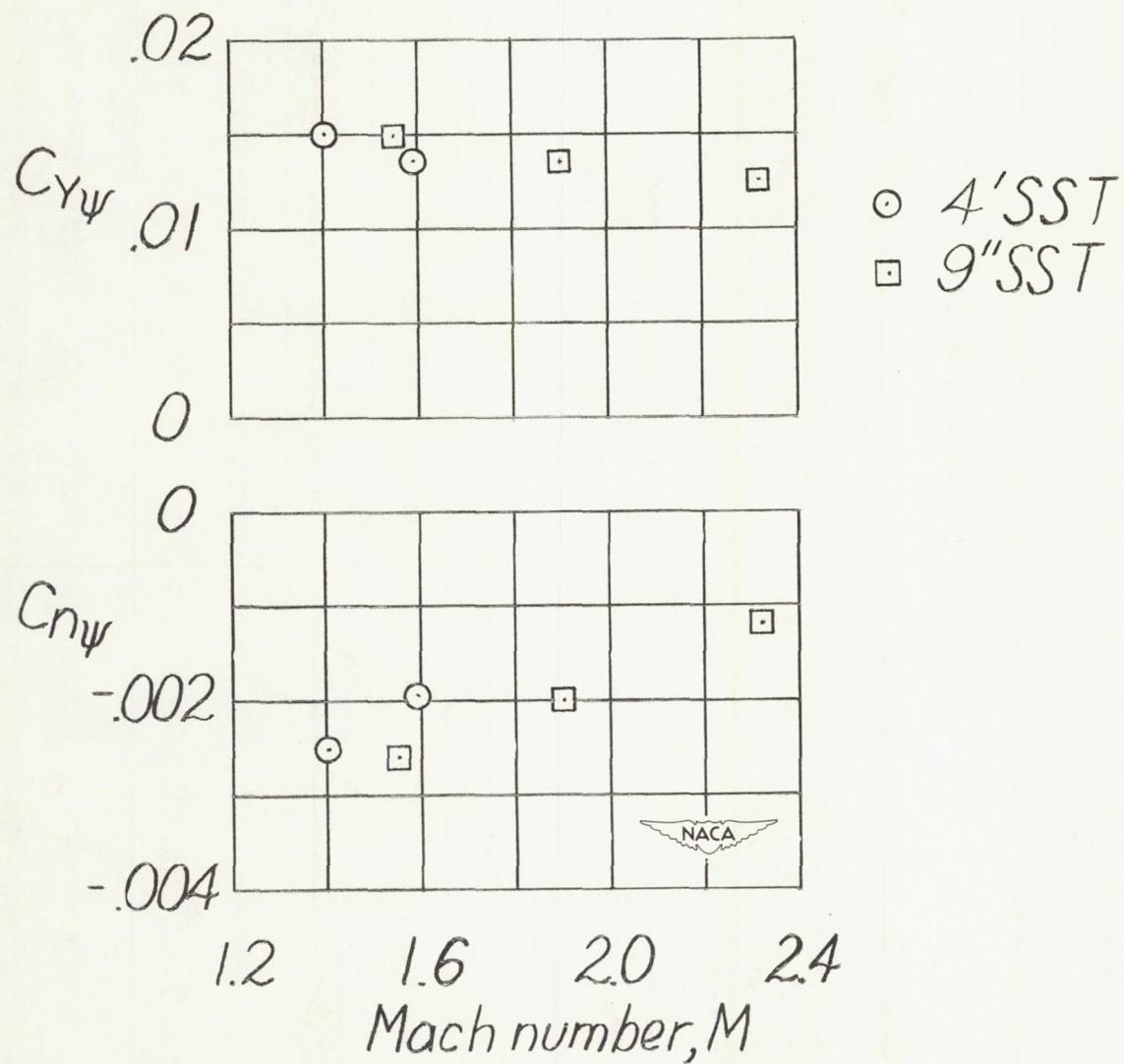


Figure 11.- Comparison of lateral-stability parameters with results from the Langley 9-inch supersonic tunnel. $C_L \approx 0$.